Technical Memorandum 104820

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ABSTRACT

During the flight history of the Space Shuttle Orbiter, several flights have experienced earlier than expected laminar to turbulent boundary layer transition. Such an experience has often been asymmetric in nature, which results in unplanned control requirements and often higher localized heating. Typically, it was noted that evidence of roughness elements, such as thermal protection system (TPS) gap fillers protruding above the surrounding surface, was generally a common factor for the flights experiencing the unexpected early transition.

We conducted a test in the Arnold Engineering and Development Center's Von Karman Facility Tunnel B at a Mach number of 8.0 and over a Reynolds number range of 0.6 x 10^6/ft to 3.6 x 10^6/ft to determine the effect of specific discrete roughness element heights and locations on the transition of the Orbiter boundary layer from laminar to turbulent at various Reynolds numbers. We limited angle of attack to 35° and 40°, which are most representative of the flight experience. The test used Shuttle wind tunnel model 29-O, a 0.0175-scale Orbiter model, to which discrete roughness elements of various heights were added at specific locations on the model's windward surfaces, specifically the lower fuselage and wing. Model instrumentation consisted of 55 heat flux gages distributed over the windward surfaces.

The major objectives of the test were to verify that off-design changes, such as protuberances, to the TPS could cause roughness-induced early transition consistent with flight experience and to determine, at various Reynolds numbers, the size and location of discrete roughness elements that would cause early transition. Such test data could then be used for the future development of an Orbiter discrete roughness transition model.

Test results indicated that, while a single roughness element on the windward centerline near the nose region could induce early boundary layer transition over most of the windward surface, a single roughness element off to the side of the windward centerline could result in a turbulent boundary layer occurring on a major portion of one side of the windward surface with the other side remaining laminar. Both increasing the height of the roughness element on the windward centerline and cooling the model resulted in boundary layer transition at lower freestream Reynolds numbers.

INTRODUCTION

This report presents the results of wind tunnel test MH-11, which was conducted in the Arnold Engineering Development Center (AEDC) Von Karman Facility (VKF) Tunnel B during the week of June 5, 1995. The purpose of the test was to determine the effect of specific discrete roughness element heights and locations on the transition of the Orbiter boundary layer from laminar to turbulent at various Reynolds numbers. Three major objectives guided the planning of this test:

A) Verify that off-design changes to the Orbiter thermal protection system (TPS) can cause roughness-induced early transition consistent with flight experience.
B) Determine the size and location of discrete roughness elements that will cause early transition.
C) Use test data to develop an Orbiter discrete roughness transition model.

TEST FACILITY

The test facility, AEDC VKF Tunnel B shown in Figure 1, consists of a closed-circuit, continuous-flow, variable-density, 50-inch-diameter test section wind tunnel (Reference 1). Through the use of interchangeable axisymmetric contoured nozzles, test conditions at Mach numbers of 6 and 8 are realized. A model injection system allows the model to be removed from the test section while the tunnel remains in operation. Continuous operation of the tunnel with a range of pressure from 100 to 850 psia is possible with air supplied by the VKF main compressor plant. Stagnation temperatures sufficient to avoid air liquefaction in the test section (up to 1,350°R) are achieved from a natural gas combustion heater. Integral external water jackets cool the entire tunnel (throat, nozzle, test section, and diffuser).

TEST CONDITIONS

The test was run at a Mach number of 8.0 with Reynolds number varying from 0.6×10^6 per foot to 3.6×10^6 per foot. The model angle of attack was primarily set at 40°, with some data taken at an angle of attack of 35°.

MODEL DESCRIPTION

The model used in wind tunnel test MH-11 is designated as the Shuttle Orbiter model 29-O and is a 0.0175-scale representation of Orbiter configuration -140B. Figure 2 illustrates the nominal dimensions of the model.

Model 29-O is constructed of stainless steel and is designed to be sting-mounted in the wind tunnel. The model had previously been etched to simulate tiles over the windward surface; however, the model was modified for this test by smoothing the etched simulated tiles to a nominal height of 0.001 inches. For the purposes of this test, the vertical tail and orbital maneuvering system pods were not included as components of the complete, as tested, wind tunnel model. A total of 48 stainless steel discrete roughness element inserts were manufactured for installation in 11 locations on the model lower surface. The inserts were constructed such that each represented either a smooth surface (zero roughness element height) or a specific discrete roughness element height. Table 1 and Figure 3 define the locations of the inserts and their element heights. Figure 4 presents the geometry of a typical discrete roughness element insert. Each insert was attached to the model surface with two recessed machine screws and the recesses were filled with dental plaster. During the test, the model was configured either with only smooth inserts or with smooth inserts in combination with either one or two discrete roughness element inserts.

MODEL NOMENCLATURE

The wind tunnel model 29-O dimensions nomenclature is as follows:

b	Wing span, 16.4 inches
BL	Y_0 coordinate in model scale (Buttock Line)
L	Reference body length, 22.58 inches
MS	X_0 coordinate in model scale (Model Station)
X	Longitudinal body axis
X_0	Full-scale Orbiter longitudinal axis coordinate
Y_0	Full-scale Orbiter lateral axis coordinate

The model 29-O component nomenclature is as follows:

B17	Fuselage body
C7	Canopy
F5	Baseline body flap
W103	Wing
P	Discrete roughness element insert

INSTRUMENTATION

Model Instrumentation

The model was instrumented with 55 coaxial thermocouple or heat flux gages and 4 standard thermocouple gages. The coaxial gages were used to measure the surface heat flux and were made from Constantan wire with a Chromel outer jacket, Figure 5. The gages were 0.0625 inches in diameter. Since the pre-drilled gage locations on the model were 0.125 inches in diameter, 0.123-inch-diameter Chromel sleeves were press fit on the gages. In order to inhibit extraneous thermoelectric emf generation, which can cause errors of up to 25%, as noted by Carl Kidd, et al. (Reference 2), the gages were cemented into the 55 predrilled 0.125-inch holes in the model using Ultrabond 552, a ceramic adhesive. This adhesive has well known high-temperature capabilities; however, for this test, because of the -320°F wall temperature requirement, we performed installation checkouts exposing the adhesive to the LN_2 with no issues encountered. After the gages were cemented into the model, they were smoothed down flush to the model surface to within a ±0.001-inch tolerance. This smoothing, accomplished by abrading the surface with a light sandpaper, formed the thermocouple junction. Table 2 and Figure 6 define the gage locations.

Four standard thermocouples, Chromel-Alumel, were mounted inside the model to monitor the internal temperatures of the model as a check for model temperature uniformity before each run. The thermocouples were designated as TMOD1 through TMOD4 and were placed on the inside surface of the windward side of the model, along the centerline between coaxial gages 8 and 10, 16 and 29, 37 and 38, and 43 and 46, respectively.

The gages were sampled at 20HZ from start of insertion to 3 seconds after the model reached the test section centerline. However, for data processing, only the data obtained after the first second on centerline were used. We reduced the data using the assumption of a homogeneous, one-dimensional, semi-infinite solid to obtain the heat flux. A previous calibration of the gage material lumped thermal parameter, $\rho C_p k^{1/2}$, i.e., $(ROCK)^{1/2}$, was used for the temperature range tested (Reference 3). The maximum heat flux uncertainty was estimated to be 9% and 12% for hot and cold wall, respectively, for heat flux greater than 5 BTU/ft^2-sec.

Facility Measurements

The AEDC Facility provided measurements of stagnation pressure and stagnation temperature. These measurements were stated to be accurate to within ±0.5% and were used to obtain freestream parameters of Mach number and Reynolds number. The accuracy of Mach number and Reynolds number was estimated to be ±0.3% and ±1.2%, respectively.

Additional qualitative visual data was provided through the use of a shadowgraph system and oil flows. The shadowgraph system obtained a standard image of the 17-inch window of the test section using 70mm film with a 1-microsecond exposure time. The resolution of the film was 250 lines/inch so that details of the boundary layer could be observed. For the oil flows, the model was painted with metal layout bluing (Dykem), and a silicone oil tinted white with titanium oxide was applied to observe the surface flow streamlines. Videotape coverage was also provided of (1) both forward and aft shadowgraph windows, (2) both top and side direct views of the model, and (3) selected views of the model during cool down in the injection tank.

LN_2 COOLING FIXTURE

Fixture Description

AEDC facility personnel designed and constructed a LN_2 cooling system for the purpose of cooling the model lower surface (windward surface) to -320°F before injecting the model into the tunnel test section. The system consisted of an A-shaped manifold, Figure 7, located less than two inches from the model surface, from which an LN_2 spray was directed toward the model surface by a series of holes in the manifold. A truck reservoir trailer located outside the test facility building supplied the LN_2, through an insulated line to a series of valves located outside the injection tank. While these valves were used to control the flow of LN_2 to the spray manifold, the system was designed so that gaseous nitrogen (GN_2) could also be supplied to warm the model and remove frost buildup and to provide a means to pressurize the injection tank. To facilitate the use of the manifold assembly in some future test, the manifold assembly has been packaged with model 29-O, which has been sealed in a special protective packaging utilized for those Shuttle models categorized as "1*" (the highest category of Shuttle models which are preserved for possible future use), and is stored at the Rockwell International facility in Downey, California.

LN_2 Operating Procedures

One of the issues with cooling the model to -320°F was the frost production on the model. To prevent excessive frost buildup, which affected the transition characteristics of the flow over the model, a set of procedures was developed and modified during the test. These procedures also were aimed at minimizing the amount of time (i.e. minimize the test cost) it took to cool the model to the desired cold wall condition. The procedures were arranged according to what segment of the test was being run and are listed below.

A) Before cold model runs

- A1) After the last warm model run, begin flowing LN_2 to the injection tank. This conditions the LN_2 line from the truck reservoir trailer to the injection tank.
- A2) Conduct general inspection of model or model change.

B) Continue from above, as well as in between cold model runs

- B1) At 30 psia before reaching the total pressure test condition:
  - B1a) Start GN_2 flow through the injection tank door nozzle to fill the tank with an N_2 atmosphere **.
  - B1b) Insert model into tunnel to defrost for 6 seconds and remove.
- B2) Increase injection tank pressure and maintain above the triple point of nitrogen (2 psia) and cycle between 2-4 psia to inhibit icing of the LN_2 spray.
- B3) Start the LN_2 spray on the model.
- B4) Stop the GN_2 flow and wait for the model to complete the cool down with the LN_2 spray.
- B5) Once model temperatures have stabilized at the desired levels, stop the LN_2 spray.
- B6) Rotate the model to the proper angle of attack, if required.
- B7) Reduce the injection tank pressure to equalize the pressure across the tunnel door before opening.
- B8) Inject the model into the tunnel.

C) Repeat runs or test continuation

C1) If a repeat run or a run at a new model attitude, but same condition is needed, return to step B1b.

C2) For a change of conditions, return to step B1a.

D) Model warm-up

D1) Insert model into test section for maximum of 10 seconds' duration. Several injections may be required before the model temperatures are stabilized at the desired level.

** Note that the start of the GN_2 flow was delayed until the model was out of the tunnel and the tunnel floor doors were closed, due to flow disturbances in the test section.

DATA REDUCTION

Tunnel Conditions

We computed the tunnel test section freestream parameters using the measured stilling chamber pressure and temperature, and the calibrated test section Mach number. Modifications to account for real gas effects were applied to the equations for a perfect gas isentropic expansion from the stilling chamber to the test section.

Data Nomenclature

Symbol	Definition
M_{inf}	Test section Mach number
P_{inf}	Test section static pressure (psia)
P_0	Test section stagnation pressure (psia)
q_{inf}	Test section dynamic pressure (psf)
Re_{ft}	Reynolds number per foot
Re_L	Reynolds number based on model length
T_{inf}	Test section gas temperature (°R)
ρ_{inf}	Test section gas density (slugs/ft^3)
μ_{inf}	Test section gas viscosity (lbf - sec/ft^2)
V_{inf}	Test section velocity (ft/sec)
T_0	Test section stagnation temperature (°R)
α	Model angle of attack (degrees)
G_i (i=1,...,55)	Model coaxial thermocouple gage identification number

X/L_i (i=1,...,55)	Axial location of model coaxial thermocouple gage
$2Y/B_i$ (i=1,...,55)	Lateral location of model coaxial thermocouple gage
Tw_i (i=1,...,55)	Model surface temperatures at fifty-five instrumentation locations (°R)
$\dot{q}_i$ (i=1,...,55)	Coaxial thermocouple gage output in engineering units (BTU/ft^2 - sec)
h_i (i=1,...,55)	Local heat transfer coefficients based on r = 1.0
$h_{i\ (0.9)}$	Local heat transfer coefficients based on r = 0.9
h_{ref}	Stagnation-point reference heat transfer coefficient based on 1-ft scaled radius (Fay-Riddell); see Section 5.5
h_i/h_{ref}	Heat transfer coefficient ratio based on r = 1.0
$h_{i\ (0.9)}/h_{ref}$	Heat transfer coefficient ratio based on r = 0.9

Tabulated Values

The quantities listed above in Data Nomenclature were tabulated for each time the data was sampled. The heat transfer coefficients are presented with the application of a recovery factor of 1.0 and 0.9.

Plotted Data

During the test, at the end of each run, the following plots were electronically displayed for review:

- Plot of windward centerline theoretical (laminar), Reference 4, and data-derived heat transfer coefficients, as derived from gages # 3, 6, 7, 8 10, 16, 20, 23, 30, 31, 33, 34, 37, 38, 40, 41, 43, 46, and 49. The theoretical heat transfer coefficients, as provided to AEDC by NASA-JSC, are shown in Table 3. An example plot is shown in Figure 8.

- Plot of span-wise theoretical (laminar) and data-derived heat transfer coefficients at an X/L of 0.3 for the gages # 23, 24, 25, 54, 87, 88, and 89; an X/L of 0.5 for gages # 34, 35, 53, 93, and 94; and at an X/L of 0.8 for gages # 45, 56, 66, 70, 75, and 101. Examples of those plots are shown in Figure 8. The span-wise theoretical heat transfer coefficients as provided to AEDC by NAS-JSC, are shown in Table 4.

ASCII Data Files

An ASCII electronic data file of the quantities listed in Data Nomenclature was generated for each time the data was sampled.

At the end of each test day, the ASCII files for each test run completed were transmitted electronically to NASA-JSC/LESC in Houston, Texas.

Equations and Methods

As described in Reference 1, the coax gage provides a measurement of the surface temperature of the model, which is assumed to be a homogenous, one-dimensional, semi-infinite solid. For this reason it is important that the thermophysical properties of the gage parts and the test panel material be closely matched. The coax gage heat flux at each instrumented location was computed for each time point (t_n) from the measured surface temperature by the following equation derived from semi-infinite solid considerations **(**Reference 5).

$$QDOT(t_n) = \frac{2(ROCK)^{1/2}}{(\pi)^{1/2}} \sum_{j=1}^{n} \frac{TW(t_j) - TW(t_{j-1})}{\sqrt{t_n - t_j} + \sqrt{t_j - t_{j-1}}} \quad , BTU / ft^2 \text{ - sec}$$

where

j = the jth term in summation,
n = the end point of summation,
t = the elapsed time from liftoff at the start of the inject cycle, sec.

The coax gage surface temperature, $TW(t_j)$, was computed by converting the millivolt output to temperature by using a curve fit of the Thermocouple Reference Tables published by the National Institute of Standards and Technology. The temperature dependence of the lumped thermal properties, $(ROCK)^{1/2}$ for the coax gages was included by means of the following equation

$$(ROCK)^{1/2} = 5.26x10^{-4}\left[\frac{TW(t_n) + TW(t_i)}{2}\right] + 0.12688, \quad BTU / ft^2 \text{ - } {}^\circ R \text{ - sec}^{1/2}$$

where

TW = the model surface temperature, ° R,
t_i = the initial time at liftoff, sec.

To reduce the effects of any electrical noise in the gage output, the values of QDOT were averaged for fifteen consecutive readings after the test article reached the tunnel centerline. The gage surface temperature was also averaged over the same interval.

The heat transfer coefficient for each gage was computed from

$$H(TT) = \frac{QDOT}{(TT - TW)} \quad , BTU / ft^2 \text{ - sec - } {}^\circ R$$

where

TT = the total temperature.

Since the actual value of recovery temperature is not known at each gage location, we calculated a second value of heat transfer coefficient as follows:

$$H(.9TT) = \frac{QDOT}{(.9TT - TW)} \quad , BTU / ft^2 \text{ - sec - } {}^\circ R$$

to show the variation over this range of recovery temperature.

In evaluating the Orbiter windward side heating rates, a normalized value must be obtained. We calculated this value using the stagnation point heating to a reference sphere.

One method of calculating the equilibrium stagnation point heat transfer coefficient incorporates the work of Fay and Riddell (Reference 6). This method is widely accepted and has been used in this work:

$$h_{ref} = \frac{8.17173\left(P_{O_2}\right)^{0.5}\left(\mu_O\right)^{0.4}\left(1 - \frac{P_{\inf}}{P_{O_2}}\right)^{0.25}\left(0.2235 + 1.35\times 10^{-5}(T_O + 560)\right)}{\left(R_N\right)^{0.5}\left(T_O\right)^{0.15}}$$

where

h_{ref} = Reference heat transfer coefficient based on Fay & Ridell theory $\left(\frac{Btu}{ft^2 \cdot \sec \cdot {}^\circ R}\right)$

P_{O_2} = Stagnation pressure *downstream* of a normal shock (psia)

μ_O = Viscosity based on stagnation temperature $\left(\frac{lb_f \cdot \sec}{ft^2}\right)$

$P_{\inf}$ = Freestream static pressure (psia)

T_O = Freestream stagnation temperature $({}^\circ R)$

R_N = Nose Radius (ft) (R_N = 0.0175 ft.)

This calculation must be performed in relation to the given freestream conditions. The computed reference value is then divided into the obtained Orbiter windward surface heating values $\left(h_{local}\right)$ for normalization.

To calculate the freestream air viscosity based on stagnation temperature, the Sutherland Law is used (Reference 7):

$$\mu_O = 2.270\frac{T_O^{3/2}}{T_O + 198.6}\times 10^{-8}\,\frac{lb\cdot\sec}{ft^2}$$

To calculate the stagnation pressure downstream of a normal shock $\left(P_{O_2}\right)$, one must know specific values for:

k = Ratio of specific heats $\left(C_p / C_v\right)$ (Assumed k = 1.40)
M_1 = Mach number *upstream* of a normal shock (same as $M_{\inf}$)
M_2 = Mach number *downstream* of a normal shock

With k and M_1 known, the Mach number downstream (M_2) of a normal shock can be found by the relation:

$$M_2 = \sqrt{\frac{\left(M_1\right)^2 + \frac{2}{k-1}}{\left(\frac{2k}{k-1}\right)\left(M_1\right)^2 - 1}}$$

Now, the following relation can be used to calculate the stagnation pressure downstream of a normal shock (Reference 8):

$$\frac{P_{O_2}}{P_{O_1}} = \frac{M_1}{M_2}\left(\frac{1+\frac{k-1}{2}\left(M_2\right)^2}{1+\frac{k-1}{2}\left(M_1\right)^2}\right)^{\left(\frac{k+1}{2(k-1)}\right)}$$

where

P_{O_1} = Stagnation pressure *upstream* of a normal shock (same as the freestream stagnation pressure - P_O)

Once these values are obtained, h_{ref} can be calculated and used to non-dimensionalize the corresponding heat transfer coefficient data.

Given local heat rate information, a value for the local heat transfer coefficient is computed as follows:

$$h_{local} = \left(\frac{\dot{q}_{local}}{r \cdot T_O - T_W}\right)$$

where

h_{local} = Local heat transfer coefficient $\left(\frac{Btu}{ft^2 \cdot \sec \cdot {}^\circ R}\right)$
$\dot{q}_{local}$ = Local heating rate $\left(\frac{Btu}{ft^2 \cdot \sec}\right)$
r = Recovery factor (assumed r = 1.0 or 0.9)
T_W = Surface wall temperature (° R)

Finally, h_{ref} is divided into h_{local} to achieve non-dimensionalized heat transfer information.

$$\left(h_{local} \big/ h_{ref} \right)$$

Average Surface Temperature

The ratio of the model temperature, T_W, to the tunnel stilling chamber temperature, T_T, necessitated the calculation of an average wall temperature. We obtained this average wall temperature by averaging the values of T_W for gages 16, 23, and 31. This average value was determined based upon the conditions at the beginning of the injection sequence, before the model had entered the air stream.

Measurement Uncertainties

Instrumentation calibrations and data uncertainty estimates were derived using methods described in Reference 9. Measurement uncertainty consists of a combination of bias and precision errors which are defined as:

$$U = \pm(B + t_{95}S)$$

where B is the bias limit, S is the sample standard deviation, and t_{95} is the 95th percentile point for the two-tailed Student's "t" distribution (95% confidence interval) which for sample sizes greater than 30 is taken equal to 2.

The estimates of the measured data uncertainties for this test, as shown in Reference 1, are presented in Table 5a. The data uncertainties for the measurements are determined from in-place calibrations through the data recording system and data reduction program.

Propagation of the bias and precision errors of measured data through the calculated data was made in accordance with Reference 7, and the results are presented in Table 5b.

OPERATIONS

Installation of the model 29-O, with a sting prebend of 30 degrees, in the AEDC Tunnel B was typically as illustrated in Figure 9 for all testing, except for the oil flow runs. Oil flow runs were accomplished with the model inverted in the tunnel to facilitate photographing the model from a window in the top of the tunnel during the runs. Typically at the VKF Tunnel B, the test model is mounted on the sting support mechanism in the installation/injection tank directly beneath the tunnel test section. This tank is separated from the tunnel by a pair of fairing doors and a safety door. When closed, the fairing doors, except for the slot for the pitch sector, cover the opening to the tank and the safety door seals the tunnel from the tank area. After the model is prepared for a data run, the personnel access door to the injection tank is closed, the tank is vented to the tunnel pressure, the safety and fairing doors are opened, the model is injected into the air stream, and the fairing doors are closed. However, due to the limited time over which data could be taken, the data were taken without closing the fairing and safety doors. The time required to close the doors would have allowed the model to warm up and violate the uniform temperature assumption. After the data are obtained, the model is retracted into the tank, the fairing and safety doors are closed and the procedures initiated for either a model change or preparation for the next run by either warming or cooling the model. Each injection cycle is termed as a run, and all data obtained are identified in the data tabulations by that run number.

We established the temperature of the windward surface of the model in the injection tank before each run, as previously described. The real-time temperature measurements for selected coaxial gages and the internal model thermocouples were displayed on a video screen to establish when the model reached the desired conditions.

We accomplished the model attitude positioning with the model in the installation/injection tank. As soon as the desired surface temperature was established, the model attitude was set and the model injected into the flow.

The data system was started at first motion of the model, while the model was still in the tank. The time required for the data system to record a single loop of data for each item being scanned, 0.065 seconds, was defined as the data loop period. This process resulted in approximately 15 data loops/second. Both the shadowgraph and the direct videos were initiated at the start of the injection cycle. A single photograph was taken by the 70-mm sequence camera for each test point, after the model was on the tunnel centerline.

For this test, runs were made at a freestream Mach number of 8.0 and Reynolds numbers from 0.62×10^6 per foot to 3.77×10^6 per foot. The angle of attack was primarily set at 40°, with sideslip set to 0°. A limited number of runs were made at an angle of attack of 35°. The most efficient method to accomplish the test runs was determined using the length of time required to cool or warm the model versus changing Reynolds number. We found it more efficient to run the hot wall runs as a series while changing only Reynolds number and then running the cold wall runs as a series while changing Reynolds number. We also found that the smoothness of the model was very critical for a boundary layer test, as one might suspect. Filling of screw holes and verifying heights of thermocouples were extremely important.

The testing consisted of a combination of hot wall test runs (no cooling of the model before injection into the tunnel) and cold wall test runs (model cooled to -320° F before injection into the tunnel). The model was cooled by liquid nitrogen, LN_2, applied to the model windward surface through the use of the LN_2 manifold system, which the AEDC facility personnel designed. The LN_2 cooling caused some problems which affected the test procedures and results. One example involved the dental plaster used to fill the screw recesses. We determined that the model must be near room temperature before applying the plaster, or, following the application of the LN_2 to the model surface, the plaster would tend to either partially or totally fall out during a run. To quickly heat the model to near-room temperature following a series of cooled model runs, we determined that the model should be injected back into the tunnel while the internal thermocouples were monitored. This usually involved several injections before the temperatures stabilized. For future consideration in a test such as this one, an internal heating element would be beneficial to assist in warming the model in preparation for model changes. We also noticed that frost buildup from the LN_2 cooling process could affect the test results by providing an effective trip to the boundary layer. Again, injection into the tunnel for a short warming period generally solved this problem.

The pretest test matrix was designed to consist of 11 test series corresponding to the configuration of the discrete roughness elements on the model. Each run was to be assigned an alphanumeric designation that would consist of the series identification and run number within the test series. That basic format was

sXrY

where the *X* after the letter *s* provides the series number and the *Y* after the letter *r* provides the run number within the series. For each test series, an alphabetic character was to be added to the test series number (ex.: s2ar3) to distinguish between sets of runs with different roughness element heights or to changes in angle of attack.

Once the test was begun, we quickly determined that it was most efficient to first run all hot wall runs and then all cold wall runs for a given configuration, varying the Reynolds number for each. While still adhering to the test series designation described above, run numbers were also recorded by the usual numeric designation.

Table 6 represents the complete "As-Run Test Matrix." Each column details the various parameters of the individual run. Column 1 defines the AEDC, or numeric, run number. Column 2 provides the Series number, as was described above. Column 3 is the punch code that is similar to the Series number. Column 4 defines the roughness element used, where an N/A is used to denote the model was smooth with no roughness element. Columns 5 through 10 provide the test conditions of Mach number, total pressure, total temperature, angle of attack, Reynolds number, and ratio of model wall temperature to total temperature. Column 11 denotes whether the cooling system was used. Columns 12 and 13 provide the date and time of the run. Column 14 indicates the delta time between that run and the preceding run, and column 15 contains various remarks that are provided for some of the runs. Note that of the 167 runs conducted during this test, 14 were repeat runs. In the interest of providing as complete a record as possible for future utilization of the results from this test, Table 7 presents the complete test log as recorded by Mr. Jeff Bennett, the Rockwell International test representative.

Near the end of the planned test series, the model was treated with metal layout bluing, Dykem, to provide a dark background in preparation for oil flow runs involving the use of a white titanium oxide and oil mixture. During the runs, the LN_2 system manifold was removed from the work bay and the model was rolled 180° and injected into the flow inverted to facilitate photographing the windward surface with a 70-mm sequence camera mounted in a window on the top of the tunnel test section. The sequence camera took photographs at 2-sec intervals during each oil flow run. The model was cleaned and a fresh application of oil was applied before the next run. The Dykem bluing tended to not remain on the model after the first attempted oil flow run. While several patterns of oil application were utilized in an attempt to provide the best results possible, the oil was so heavily applied that the resulting oil streamline patterns were adversely affected for most runs.

To assist in documenting all aspects of this test for future use, photographs were taken of the only original model components that were available during the pretest planning. As illustrated in Figures 10 and 11, all that remained of the original model 29-O were the wing/lower fuselage, canopy block, and associated upper wing cover plates.

Installation photographs were taken (Figure 9 and Figure 12, respectively) to document the LN_2 manifold system and its relationship to the model while the model was in the installation/injection tank beneath the test section and of the model positioned for a typical run in the test section.

During the test, multiple shadowgraphs were taken for each run. These were a post-test product which was delivered as a set of 70-mm negatives and a corresponding set of contact prints.

DATA PRESENTATION AND RESULTS

Overview of Test Data

We conducted a total of 167 runs during this test program. Of these, 5 runs (# 145 - 149) were oil flow runs and no heat flux data were obtained. For 27 runs (#1-13, 38, 41, 43, 46, 47, 49, 55, 82, 91, 115, 127, 136, 166, 167), data were not obtained or the data were corrupted due to unintentional roughness. For 18 runs (# 15, 17, 19, 22, 34, 86, 101-108, 130-132, 135) excessive frost was observed on the model due to the LN_2 cooling. Therefore, we obtained good heat flux data that can be used to evaluate the desired transition effects for 117 runs. Table 8 presents a summary list of these runs organized by configuration (smooth body or roughness element installed). Table 9 presents those runs which were deemed faulty and were not used.

Shock Interference Effect

Initial runs of the test matrix exhibited unexpected, repeatable turbulent-level heating past the X/L = 0.75 location. We believed this phenomenon to be due to transition on the model when the higher Reynolds numbers were run. However, the freestream conditions were decreased such that laminar flow was expected and yet, the heating at the aft of the model continued to be high. We realized that something was forcing the heating to approach turbulent levels at the rear of the model regardless of the freestream conditions.

As part of the regular matrix of conditions, we planned a set of lower angle of attack, as well as inverted model, test points. Upon lowering the angle of attack to 35°, the heating at the aft end of the model for the lower Reynolds numbers approached laminar levels. The same was seen for both the 40° and 35° cases when the model was inverted, thereby verifying that something was forcing the boundary layer to be turbulent toward the aft end of the vehicle for 40° angle of attack in the non-inverted configuration.

Investigation of the temperature time history plots for a number of centerline thermocouples revealed that, as the model was approaching the centerline of the tunnel, the thermocouples indicated a step increase in temperature sequentially down the vehicle for both angles of attack, and for both the inverted and normal model orientations. This effect was attributed to a stationary shock that the model passed through during insertion into the tunnel, which would also cross each thermocouple sequentially from nose to tail.

We investigated two plausible explanations for the existence of a shock in the tunnel. The first involved the reflection of a shock wave from the pit beneath the model. The second explanation was the formation of a tunnel wall boundary layer induced shock upstream of the model. Shock reflection from the pit was deemed unlikely because of the model positioning with respect to the tunnel, and the effect seemed to be stationary as the model approached the tunnel centerline. The formation of a boundary layer induced shock could be explained, however. Since the heat transfer tests being performed had very short on centerline times, the insertion doors were not being closed during a data point. The process for a model insertion involved opening the insertion doors, inserting the model for the required duration, returning the model to the pit, and finally closing the insertion doors. The fact that the insertion doors remained open during the time the model was on centerline opened the possibility to a disturbance in the pit feeding forward in the tunnel wall boundary layer. The most likely scenario involved the formation of a separation bubble pair in the pit, and in the upstream tunnel boundary layer. The formation of a small separation bubble in the upstream boundary layer on the tunnel wall would cause the formation of a shock emanating from the beginning of the separation bubble.

A tunnel survey of the Mach 8 test section flow field had been performed in August and September of 1991; however, no data were taken with the insertion doors open. A similar survey of the Mach 6 test section was performed with the insertion doors open and closed in 1974, and the existence of a tunnel wall boundary layer induced shock was documented for the open-door configuration (Reference 10). This Mach 6 tunnel survey identified a disturbance in the flow field which appears to originate ahead of the open cavity door (Figure 13). The angle of the disturbance relative to the tunnel centerline can be expected to differ in the Mach 8 facility. Although similar data for the Mach 8 facility does not exist, the results of this test demonstrate the existence of a similar disturbance. The Mach 6 test section survey thereby validates the aforementioned explanation of the higher than expected temperatures on the aft end of the model, and the existence of a disturbance like that shown in Figure 13 would explain the spatially sequential, step increase in thermocouple readings during model insertion.

The correction for runs after run 107 involved repositioning the model in the tunnel, above the location of the boundary layer induced shock. Figures 14 and 15 illustrate the original and revised model positions. Table 8 includes a column that indicates whether this problem was present during a particular run.

Smooth Body Transition Data

Effect of Reynolds Number

During this test program, test data were obtained for freestream Reynolds numbers from 0.6×10^6 per foot to 3.6×10^6 per foot. Figure 16 presents the effect of the variation of the freestream Reynolds number on the smooth body transition for the centerline of the model. Recall that the smooth body included a distributed roughness of 0.001 in. on the windward surface. At the highest Reynolds number (3.58×10^6 per foot), the transition front is located at approximately X/L = 0.4. As the freestream Reynolds number was decreased, the transition front moved further back on the model toward the tail. At Reynolds number = 1.52×10^6 per foot, we observed mostly laminar flow on the model. However, near the tail end of the model there is a slight increase in heating that may be due to near-transitional heating. Figures A-1 through A-5 in Appendix A provide comparisons of the measured heat transfer coefficients to the theoretical laminar values for the entire windward surface. The base color contours represent the predicted laminar heating, and the colors within the circles correspond to the heating levels measured by the heat flux gages. When the colors within the circles nearly match the surrounding color, then laminar flow is indicated. When the color within the circle indicates higher heating levels than the surrounding colors, then transitional to turbulent heating levels are indicated. These plots provide a more complete description of the transition front on the model windward surface.

Effect of Angle of Attack

Figure 17 presents the centerline heat transfer data for two Reynolds numbers (3.58×10^6 and 1.52×10^6 per foot) at 35° and 40° angle of attack. The centerline data indicate that an angle-of-attack change from 35° to 40° has little affect on the transition location and both cases appear to experience transitional heating at the tail-end of the model for the lower Reynolds number. However, the color contour plots presented in Figures A-1, A-5, A-6, and A-7 in Appendix A indicate that the transition front has moved further forward on the wing for the 40° case than for the 35° case.

Effect of Wall Cooling

Figure 18 presents the centerline heat transfer data for two Reynolds numbers (3.58×10^6 and 2.50×10^6 per foot) and two average wall temperature ratios (T_W/T_O=0.37 and T_W/T_O=0.13). At the higher Reynolds number, the cold wall case results in a significant jump forward of the transition front to approximately X/L = 0.2 as compared to X/L = 0.4 for the warm wall case. Furthermore, for the cold wall case, the transition process from laminar to turbulent occurs across a shorter distance. For the lower Reynolds number, 2.5×10^6 per foot, the transition front is at approximately X/L = 0.55 for the warm wall and at X/L = 0.35 for the cold wall. Again, turbulent levels of heating are attained more rapidly for the cold wall case. Figures A-1, A-3, A-8, and A-9 in Appendix A provide the corresponding color plots for these cases. The change in wall temperature results in a dramatic change in the transition location. At the high Reynolds number (Figure A-8), it is unknown why the transition front appears to be located further upstream on one side of the model than the other.

Roughness Element Transition Data

Effect of Reynolds Number

Figure 19 presents the resultant heat transfer distributions on the model centerline with a 0.008-inch roughness element located at Location D. At the highest Reynolds number, 2.51†x 10^6 per foot, transition occurs at the roughness element, and turbulent levels of heating are attained rapidly. By decreasing the Reynolds number to 2.05 x 10^6 per foot, transition still occurs at the element. But after an initial increase in heating, turbulent heating is attained considerably further downstream. As the Reynolds number is further decreased to 1.49 x 10^6 per foot, the initial increase in heating is less and the transition to turbulence is moved even further downstream. At a Reynolds number of 0.998 x 10^6 per foot, a small disturbance in the heating near the element is noticed but the rest of the model appears to be laminar. Figures A-5, A-10, A-11, A-12, and A-13 in Appendix A present the corresponding contour plots.

Effect of Angle of Attack

Figure 20 presents the centerline heat transfer results for a 0.015-inch roughness element located on model centerline at Location G. Three runs were made with a Reynolds number of 1.0 x 10^6 per foot for 40°, 38°, and 35° angle of attack. For all three cases, the transition location is at the element and for 40° and 38° little difference in the results are observed. However, at 35° the distance between transition onset, at the element, to full turbulence is slightly delayed. The color plots presented in Figures A-14, A-15, and A-16 in Appendix A show that the effect of the roughness element is restricted to the center area of the model downstream of the element.

Figure 21 presents the centerline results for a roughness element located off-centerline at Location C for a Reynolds number of 2.0 x 10^6 per foot. Results for two roughness heights—0.008 and 0.015 inches—and for 35° and 40° angle of attack are presented. For both heights, the roughness elements appear to have little affect on the model centerline heating at 40° angle of attack. However, at 35° the smaller roughness element results in the transition front crossing the centerline at approximately X/L = 0.5. Referring to the color plots in Figures A-17, A-18, A-19, and A-20 in Appendix A, you can see that both roughness elements have a greater influence on transition at the lower angle of attack and that the smaller roughness element has more of an affect than the larger one. These phenomena will require further investigation.

Figure 22 presents the centerline results for a 0.015-inch roughness element located off-centerline at Location F for a Reynolds number of 2.0 x 10^6 per foot for both 35° and 40° angle of attack. No affect on transition is observed on the centerline. However, Figures A-4, A-21, A-22, and A-23 in Appendix A indicate that at 35° the roughness element has more affect on transition off-centerline than for the 40° case.

Effect of Wall Cooling

Figure 23a-c compares the centerline heat flux results for a warm wall and cold wall model with a 0.008-inch roughness element at Location D. As shown in Figure 23a for a Reynolds number of 2.0 x 10^6 per foot, the wall temperature had a significant effect on transition on the smooth body. However, for the case with the roughness element, transition occurred at the element for both wall temperatures, and the rise to turbulent heating has similar trends. Similar trends are observed at a lower Reynolds number of 1.5 x 10^6 per foot (Figure 23b). At a Reynolds number of 1.0 x 10^6 per foot (Figure 23c), both the smooth model and the model with a roughness element have mostly laminar heating for the warm wall case except for a slight increase in heating just downstream of the roughness element. For the cold wall case, both model configurations have heating levels slightly higher than laminar on the aft half of the model, which may indicate transitional heating. Figures A-11, A-24, A-25, and A-26 in Appendix A provide the contour plots for a Reynolds number of 2.0 x 10^6 per foot. Figures A-12, A-27, A-28, and A-29 present the contour plots for a Reynolds number of 1.5 x 10^6 per foot. Figures A-13, A-30, A-31, and A-32 present the contour plots for a Reynolds number of 1.0 x 10^6 per foot.

Effect of Roughness Element Height

At Location A on the model, two roughness element heights were tested—0.004 and 0.008 inches. At a Reynolds number of 2.0 x 10^6 per foot (Figure 24), transition occurs on the smooth body at approximately X/L = 0.6. The small roughness element brings the transition front forward to approximately X/L = 0.4. The large roughness element causes transition to occur at the element. Figures A-4, A-33 and A-34 in Appendix A present the contour plots for these cases.

At Location G on the model, two roughness element heights were also tested—0.006 and 0.015 inches. For a Reynolds number of 1.5 x 10^6 per foot (Figure 25), the largest roughness element brings transition to the element; whereas the small roughness element has little effect on transition when compared to the smooth body results. Figures A-5, A-35, and A-36 present the contour plots.

Effect of Roughness Location

With a 0.005-inch roughness element at Location E and a 0.006-inch roughness element at Location I, little effect on transition on the model centerline was observed for a Reynolds number of 2.5 x 10^6 per foot (Figure 26). However, the color plot in Figure A-37 indicates that the roughness element at I brings transition to the element, but the element at E has no apparent effect on transition.

A 0.010-inch and a 0.008-inch roughness element were tested simultaneously at Locations J and K, respectively. As shown in Figure 27, the transition front is located at element J for a Reynolds number of 1.5 x 10^6 per foot. Reviewing the color plot in Figure A-38, element K also brings the transition front to the element. In both cases, the effect of the elements appears to be restricted to a narrow region downstream of the element.

Oil Flow Data

Oil flows were run at both 35° and 40° angle of attack. Several patterns of titanium oxide and oil mixture applications were utilized in an attempt to provide the best results possible. However, in most cases, the oil mixture was so heavily applied that the resulting oil streamline patterns were not of the best quality. The Dykem bluing, used to provide a dark surface background for the white titanium oxide and oil mixture, tended to not remain on the model after the first oil flow attempt. The last three oil flow runs (147, 148, and 149) were of an acceptable quality. Additionally, roughness element C-4 was installed on the model for these runs. The results of runs 147 and 148 at angles of attack of 35° and 40° are presented in Figures 28 and 29, respectively. The higher shear due to turbulent flow on the model's right side is evident in the photos.

Shadowgraph Data

During each test run, a shadowgraph was obtained of the Orbiter model in the test section. These profile photographs show clearly the bow shock around the model. Details of the boundary layer state—laminar or turbulent—are not discernible from the photographs. Due to the sheer number of photographs involved, only Figures 30 and 31 (Runs 27 and 37) are presented as examples of the shadowgraphs taken. Run 27 was a smooth-body run and the boundary-layer was mostly laminar. During Run 37, the model had a 0.008-inch roughness element on the model centerline at $X/L \sim 0.05$ and resulted in a turbulent boundary layer for most of the body.

Plotted and Tabulated Data

Appendix B and Appendix C contain the plotted and tabulated data for the test. Data from runs 2, 3, 11, 12, and 38 were not recovered. Appendix B, the plotted data, includes both the data obtained along the model centerline and the span-wise distributions at three axial locations, $X/L = 0.3$, 0.5, and 0.8. The plotted heat flux gage data are compared to the computed laminar heating levels.

CONCLUSIONS

Heat flux data have been obtained on the windward surface of a 0.0175-scale Orbiter model at a Mach number of 8 and over a Reynolds number range of 0.6×10^6 to 3.6×10^6/ft. Single rectangular roughness elements were positioned on the model at nine different locations to evaluate the effect of discrete roughness on the boundary layer transition. From the test results, the following conclusions can be drawn:

A) A single roughness element on the windward centerline in the nose region can induce early boundary layer transition over most of the windward surface.

B) A single roughness element off to the side of the windward centerline can result in a turbulent boundary layer occurring on a major portion of one side of the windward surface while the other side remains laminar.

C) Small changes in angle of attack (5 degrees) have little effect on transition for a smooth body or for roughness elements located on the windward centerline. However, for off-centerline roughness elements, the transition trends are affected by angle of attack.

D) Cooling the model results in boundary layer transition occurring at lower freestream Reynolds numbers.

E) Increasing the height of roughness elements located on the windward centerline results in transition at lower freestream Reynolds numbers. However, the trend is less obvious for the off-centerline elements.

F) Roughness elements located further aft on the model result in only narrow regions of turbulence on the model.

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